National Advisory Committee for Aeronautics

Research Abstracts

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CURRENT NACA REPORTS

NACA Rept. 1096

EXPERIMENTAL DETERMINATION OF THE EFFECT OF HORIZONTAL-TAIL SIZE, TAIL LENGTH, AND VERTICAL LOCATION ON LOW-SPEED STATIC LONGITUDINAL STABILITY AND DAMPING IN PITCH OF A MODEL HAVING 45° SWEPTBACK WING AND TAIL SURFACES. Jacob H. Lichtenstein. 1952. ii, 22p. diagrs., photos., 3 tabs. (NACA Rept. 1096. Formerly TN 2381; TN 2382)

An investigation has been conducted in the Langley stability tunnel to determine the effects of horizontal tails of various sizes and at various tail lengths (when located on the fuselage center line) and also the effects of vertical location of the horizontal tail relative to the wing on the low-speed static longitudinal stability and on the steady-state rotary damping in pitch for a complete-model configuration. The wing and tail surfaces had the quarter-chord lines swept back 45° and had aspect ratios of 4.

NACA Rept. 1098

SUMMARY OF METHODS FOR CALCULATING DYNAMIC LATERAL STABILITY AND RESPONSE AND FOR ESTIMATING LATERAL STABILITY DERIVATIVES. John P. Campbell and Marion O. McKinney. 1952. ii, 40p. diagrs., 2 tabs. (NACA Rept. 1098. Formerly TN 2409)

A summary of methods for making dynamic lateral stability and response calculations and for estimating the aerodynamic stability derivatives required for use in these calculations is presented. The processes of performing calculations of the time histories of lateral motions, of the period and damping of these motions, and of the lateral stability boundaries are presented as a series of simple straightforward steps. Existing methods for estimating the stability derivatives are summarized and, in some cases, simple new empirical formulas are presented. Detailed estimation methods are presented for low-subsonic-speed conditions but only a brief discussion and a list of references are given for transonic- and supersonic-speed conditions.

NACA Rept. 1102

THE LINEARIZED CHARACTERISTICS METHOD AND ITS APPLICATION TO PRACTICAL NONLINEAR SUPERSONIC PROBLEMS. Antonio Ferri. 1952. ii, 18p. diagrs. (NACA Rept. 1102. Formerly TN 2515)

The method of characteristics has been linearized by assuming that the flow field can be represented as a basic flow field determined by nonlinearized methods and a linearized superposed flow field that considers small changes in boundary conditions. The method has been applied to two-dimensional rotational flow, to calculations of axially symmetric flow, to slender bodies without symmetry, and to wing problems.

NACA Rept. 1104

PRELIMINARY INVESTIGATION OF A NEW TYPE OF SUPERSONIC INLET. Antonio Ferri and Louis M. Nucci. 1952. ii, 19p. diagrs., photos., tab. (NACA Rept. 1104. Formerly TN 2286)

A supersonic inlet with supersonic deceleration of the flow entirely outside of the inlet is considered. A particular arrangement with fixed geometry having a central body with a circular annular intake is analyzed, and it is shown theoretically that this arrangement gives high pressure recovery for a large range of Mach number and mass flow and, therefore, is practical for use on supersonic airplanes and missiles. Experimental results confirming the theoretical analysis give pressure recoveries which vary from 95 percent for Mach number 1.33 to 86 percent for Mach number 2.00. These results were originally presented in a classified document of the NACA in 1946.

NACA TN 3055

FRICTION AND WEAR INVESTIGATION OF MOLYBDENUM DISULFIDE. I - EFFECT OF MOISTURE. Marshall B. Peterson and Robert L. Johnson. December 1953. 28p. diagrs., photos. (NACA TN 3055)

Low-speed kinetic friction studies show that for molybdenum disulfide MoS_2 lubricated steel surfaces coefficients of friction were greater at high humidity than with dry air. MoS_2 powder did not adhere to steel surfaces at high humidity, and as a result, metallic contact was greater and friction and wear increased. Increased wear with greater humidity may have been caused by corrosion of steel specimens by acids formed on contact of moisture with MoS_2. Variations in shear area resulted from changes in humidity, slider geometry, surface finish, and method of film application. Larger shear areas caused greater friction when MoS_2 filled the surface interstices and was sheared.

NACA TN 3058

TRANSIENT TEMPERATURE DISTRIBUTIONS IN SIMPLE CONDUCTING BODIES STEADILY HEATED THROUGH A LAMINAR BOUNDARY LAYER. Hermon M. Parker. December 1953. 42p. diagrs. (NACA TN 3058)

An analysis is made of the transient heat-conduction effects in three simple semi-infinite bodies: the insulated flat plate, the conical shell, and the slender solid cone. The bodies are assumed to have constant initial temperatures and at zero time to begin to move at a constant speed and zero angle of attack through a homogeneous atmosphere. The heat input is taken as that through a laminar boundary layer. Radiation heat transfers and transverse temperature gradients are assumed to be zero. The appropriate heat-conduction equations are solved by an iteration method, the zeroeth order terms describing the situation in the limit of small time. The method is presented and the solutions are calculated to three orders which are sufficient to give reasonably accurate results when the forward edge has attained one-half the total temperature rise (nose half-rise time). Flight Mach number and air properties occur as parameters in the result. Approximate expressions for the extent of the conduction region and nose half-rise times as functions of the parameters of the problem are presented.

NACA TN 3087

EXPERIMENTAL INVESTIGATION OF TWO-DIMENSIONAL TUNNEL-WALL INTERFERENCE AT HIGH SUBSONIC SPEEDS. Earl D. Knechtel. December 1953. 13p. diagrs. (NACA TN 3087)

Results are shown of an investigation of tunnel-wall interference in a two-dimensional-flow, rectangular, closed-throat wind tunnel through a Mach number range from 0.3 to 0.9 and a corresponding Reynolds number range from 0.9×10^6 to 1.8×10^6. For ratios of airfoil chord to tunnel height below approximately 0.15, correction of aerodynamic data for wall interference by the small-perturbation theory of NACA Rep. 782 was found to yield results in satisfactory agreement with essentially interference-free data.

NACA TN 3089

ONE-DIMENSIONAL, COMPRESSIBLE, VISCOUS FLOW RELATIONS APPLICABLE TO FLOW IN A DUCTED HELICOPTER BLADE. John R. Henry. December 1953. 16p. diagrs. (NACA TN 3089)

One-dimensional, steady-state, compressible, viscous flow relations are presented which permit the determination of flow conditions at any radial position in a ducted helicopter blade. The relations are required for estimating the performance of proposed helicopter jet-propulsion systems which involve ducting air or gases through the blade from root to tip. A limited number of calculations over a wide range of helicopter operating conditions and relative duct sizes are also presented. The "choking" problem in the straight duct is discussed.

NACA TN 3090

INVESTIGATION OF SANDWICH CONSTRUCTION UNDER LATERAL AND AXIAL LOADS. Wilhelmina D. Kroll, Leonard Mordfin and William A. Garland, National Bureau of Standards. December 1953. 58p. diagrs., photos., 4 tabs. (NACA TN 3090)

Tests under combined axial load and lateral pressure were made of sandwich panels with simply supported loaded edges and free unloaded edges to determine the strength of panels of various thicknesses and to compare the results with computed values. The theory, derived in this paper, is based on the theory for buckling of simply supported sandwich columns and was conservative in predicting larger strains or deflections than those measured. Agreement between computed maximum loads and experimental failing loads was within 9 percent.

NACA TN 3093

EFFECT OF TYPE OF POROUS SURFACE AND SUCTION VELOCITY DISTRIBUTION ON THE CHARACTERISTICS OF A 10.5-PERCENT-THICK AIRFOIL WITH AREA SUCTION. Robert E. Dannenberg and James A. Weiberg. December 1953. 59p. diagrs., photos., 5 tabs. (NACA TN 3093)

Results are presented of an investigation of a two-dimensional, 10.51-percent-thick symmetrical airfoil with area suction near the leading edge. Lift and suction-flow characteristics were determined with different porous surfaces (perforated plates and sintered steel) for various suction velocity distributions. The flow requirements were ascertained over a range of free-stream velocities. The maximum lift was independent of the surface of the materials tested.

NACA RM E53J12

EFFECT OF DIFFUSION PROCESSES AND TEMPERATURE ON SMOKING TENDENCIES OF LAMINAR DIFFUSION FLAMES. Rose L. Schalla. December 1953. 23p. diagrs., photos. (NACA RM E53J12)

The effects of diffusion processes on the smoking tendencies of eight laminar diffusion flames were investigated by varying the rate and concentration of air and oxygen supplied to the flame. Increasing the rate at which air was supplied permitted rather limited increases in the smoke-free burning rate. Increasing the concentration of oxygen from 21 to 45 percent increased the smoke-free fuel flow for pentene-1, neopentane, isobutane, ethylene, and n-butane, but decreased the smoke-free burning rate for butene-1, cyclopropane, and propene in the range of 25 to 40 percent oxygen. Increasing the flame temperature caused the smoke-free fuel flow to increase except for butene-1, cyclopropane, and propene, where initial rates were lower. The variations in smoke formation with oxygen enrichment were tentatively explained by considering the effect of temperature on the initial decomposition reactions of the fuel molecule.

BRITISH REPORTS

N-27230*

Aeronautical Research Council (Gt. Brit.)
DYNAMIC LONGITUDINAL STABILITY MEASUREMENTS ON A SINGLE-ROTOR HELICOPTER (HOVERFLY MK. I). W. Stewart. With an appendix on THE THEORETICAL ESTIMATION OF THE STABILITY. G. J. Sissingh. 1953. 24p. diagrs. (ARC R & M 2505; ARC 11,578. Formerly RAE Aero 2244)

Flight measurements have been made of the phugoid motion of the Hoverfly Mk. I helicopter, following an arbitrary longitudinal displacement of the control, the latter being returned to its initial position and held fixed. The tests were done throughout the speed range for power-on conditions and in autorotation for various center-of-gravity positions and for forward and backward initial displacement of the stick.

N-27231*

Aeronautical Research Council (Gt. Brit.)
TESTS ON THE HURRICANE L. 1696 IN THE 24-FT WIND TUNNEL. D. W. Bottle and T. V. Somerville. 1953. 10p. diagrs., photo., 5 tabs. (ARC R & M 2562; ARC 5344. Formerly RAE BA 1697)

Tests on a Hurricane in the 24-foot wind tunnel at the Royal Aircraft Establishment were required to find if any simple modifications could be made which would reduce its drag. Measurements were made of: (1) leak drag, (2) drag of miscellaneous excrescences, (3) cooling drag, and (4) drag of the tail unit. The tests showed that the leak drag plus the drag due to the control gaps was 13 percent of the total profile drag of the aircraft. Of this leak drag, only one-third could be eliminated by methods which could be incorporated in production aircraft without serious modification. This emphasizes the importance of eliminating leaks in the design stage. The drag of the cooling system was reasonably low, and tail-fuselage interference was small.

N-27232*

Aeronautical Research Council (Gt. Brit.)
LANDING OF AN AIRCRAFT ON A SUSPENDED SHEET. J. Taylor. 1953. 37p. diagrs., 3 tabs. (ARC R & M 2574; ARC 10,643. Formerly RAE Structures 3)

An investigation is made into the characteristics of a freely suspended flexible sheet as a shock absorber replacing the conventional undercarriage, particular attention being given to the inertia of the sheet. It is found that when an aircraft is dropped vertically on to the sheet the retarding force is first produced by the inertia of the sheet itself, and not until later in the descent by the reactions from the side supports of the sheet. By careful adjustments of the mass and tension of the sheet "retardation efficiencies" exceeding 80 percent can be achieved. The effect of the aircraft having a forward component of velocity increases the contribution of sheet momentum. For reasonably practical landing speeds and sheet dimensions, virtually the whole of the momentum of descent is absorbed by sheet inertia. Under such conditions still higher retardation efficiencies are obtainable and, with a suitable design of aircraft keel, rebound may be entirely eliminated.

N-27233*

Aeronautical Research Council (Gt. Brit.)
WIND-TUNNEL TESTS ON THE 30 PERCENT SYMMETRICAL GRIFFITH AEROFOIL WITH DISTRIBUTED SUCTION OVER THE NOSE. N. Gregory, W. S. Walker and A. N. Devereux. 1953. 14p. diagrs., photos. (ARC R & M 2647. Formerly ARC 11,599; Perf. 461; FM 1260)

This report describes tests carried out on the 30 percent Griffith symmetrical airfoil with continuous suction applied through a porous capping fitted over the front 15 percent of the upper surface. Throughout the range of incidence covered in the experiments, distributed suction was found to decrease the slot suction necessary to prevent separation, especially when the distributed suction caused rearward movement of the transition position. The profile drag of the airfoil was measured, and estimates were made of the equivalent drag coefficients for the work done by the suction pumps. Assuming no losses additional to those in the boundary layer, it was found that the effect of distributed suction was to reduce slightly the overall drag of the airfoil. Measurements of the velocity within the boundary layer were made at various chordwise positions on the porous surface; the profiles recorded were very close to the theoretical. Distributed suction was able to delay transition when this would otherwise be precipitated by a ridge on the surface, or by adverse pressure gradients, but a turbulent boundary layer remained turbulent when suction was applied. The characteristic spread of turbulent flow in the wake of a small particle on the surface was much reduced by distributed suction; under favorable conditions, the wake was entirely eliminated.

N-27234*

Aeronautical Research Council (Gt. Brit.)
AEROFOIL OSCILLATIONS AT HIGH MEAN INCIDENCES. W. P. Jones. 1953. 9p. diagrs. (ARC R & M 2654. Formerly ARC 11,502; 0.724)

The problem of the estimation of the aerodynamic forces acting on two-dimensional airfoils oscillating at mean incidences below the stall is considered. A method of calculation is suggested which makes use of the steady motion characteristics of the airfoil. At low frequencies, good agreement with the measured aerodynamic derivatives should be obtained as the method is such that it gives the correct values at zero frequency. A comparison between the estimated and measured values of the pitching-moment derivatives for a particular airfoil is made, and this shows that the method suggested gives better agreement with experiment than the usual vortex-sheet theory.

N-27235*

Aeronautical Research Council (Gt. Brit.)
CORRECTIONS TO VELOCITY FOR WALL CONSTRAINT IN ANY 10 x 7 RECTANGULAR SUBSONIC WIND TUNNEL. J. Y. G. Evans. 1953. 42p. diagrs., photos. (ARC R & M 2662; ARC 12,526. Formerly RAE Aero 2307)

The validity and accuracy of methods of determining corrections to the measured velocity in a wind tunnel to compensate for the constraining effect of the walls are reviewed following recent experimental evidence from the R. A. E. 10- by 7-foot subsonic wind tunnel. It is concluded that such corrections, commonly known as "blockage" corrections, can be successfully applied at Mach numbers up to 0.96 but some modifications are necessary to the formulas at present in use. Formulas for the calculation of the longitudinal distribution of blockage increment due to any model, necessary to check the validity of the method in particular cases, are presented in a form which, it is hoped, will facilitate their use in any 10 by 7 wind tunnel. Formulas for the corresponding wall velocity increments, used to check the accuracy of the method by comparison with measured wall pressures, are also given.

N-27236*

Aeronautical Research Council (Gt. Brit.)
SOME HIGH-SPEED TESTS ON TURBINE CASCADES. E. A. Bridle. 1953. 14p. diagrs., photos. (ARC R & M 2697; ARC 12,455. Formerly NGTE R. 48)

High-speed wind-tunnel tests on seven cascades of turbine blades are described, the blades having conventional sections including both reaction and impulse designs. The two-dimensional performance over wide ranges of incidence at Mach numbers up to 1.0 is discussed, special importance being attached to the effects of compressibility. It is shown that the effect of increasing the degree of reaction is to reduce the total-head loss and to increase the unstalled incidence range. High Mach numbers alone are not found to cause a catastrophic increase in loss with these particular blade designs, while the $\cos^{-1}$ throat/pitch rule is found to be approximately true only at high Mach numbers.

N-27237*

Aeronautical Research Council (Gt. Brit.)
THE IMPROVEMENT IN PRESSURE RECOVERY IN SUPERSONIC WIND TUNNELS. H. Eggink. 1953. 18p. diagrs., photos. (ARC R & M 2703; ARC 12,527. Formerly RAE Aero 2326; SD 34)

The inefficient pressure recovery of present-day supersonic wind tunnels, which leads to high costs of plant installation and operation, is discussed and methods of improvement suggested. In particular, the diffuser system, where most of the losses occur, is studied in detail; the improvement to be expected in the pressure recovery by the use of convergent-divergent types is explained and methods of overcoming the necessity for high starting powers with this arrangement are presented. Diffuser experiments based on recent investigations into breakaway phenomena in supersonic flow are described which result in a considerable improvement of pressure recovery. A deceleration from $M = 2.48$ at the working section to $M = 1.42$ at the diffuser throat was obtained using a variable diffuser throat.

N-27238*

Aeronautical Research Council (Gt. Brit.)
THE DESIGN AND TESTING OF SUPERSONIC NOZZLES. R. Harrop, P. I. F. Bright, J. Salmon and M. T. Caiger. 1953. 40p. diagrs., photos., 7 tabs. (ARC R & M 2712; ARC 12,114; ARC 13,383. Formerly RAE Aero 2293; RAE Aero 2293a)

The theory of the flow through a throat near sonic velocity is developed, and is followed by a discussion of the conventional method of designing supersonic nozzles using the method of characteristics. A method of improving the Mach number distribution of the nozzle using the experimental results is developed. The nozzles designed were tested in a 3-inch square wind tunnel in which the Mach number distribution was obtained by shaping the top wall of the working section. The Mach number distribution along the bottom wall was determined from the pressures measured by a series of static-pressure holes along the wall. Considerable difficulty was found in improving the distribution; this was considered to be due to the discontinuity in curvature at the point of inflexion and the influence on the boundary layer of the sudden relaxation of the pressure gradient along the wall. An alternative method of design was developed which avoided this discontinuity in curvature, and considerably better results were obtained when attempts were made to improve the experimental Mach number distribution.

N-27239*

Aeronautical Research Council (Gt. Brit.)
ELASTICITY OF A SHEET REINFORCED BY STRINGERS AND SKEW RIBS, WITH APPLICATIONS TO SWEPT WINGS. E. H. Mansfield. 1953. 27p. diagrs. (ARC R & M 2758; ARC 12,990. Formerly RAE Structures 52)

A rigorous theory has been developed for determining the stresses and displacements in a sheet reinforced by stringers and ribs which are not at right angles to the stringers. The solution of many problems of practical importance has been facilitated by the introduction of a stress function. The theory has been applied to a cylinder of rectangular section stiffened with such skew ribs (a simplified representation of a swept wing). It is shown that there are axes about which applied moments produce pure twist or pure curvature of the cylinder. There are simple formulas for determining these axes and the relationships between twist and curvature and the applied moments.

N-27240*

Aeronautical Research Council (Gt. Brit.)
POLYMETHYL METHACRYLATE (PERSPEX TYPE) PLASTICS; CRAZING, THERMAL AND MECHANICAL PROPERTIES. H. Warburton Hall and E. W. Russell. 1953. 27p. diagrs., photos., 2 tabs. (ARC R & M 2764; ARC 12,829. Formerly RAE Chem. 454)

The report summarizes the more practical aspects of the results of a long-term investigation of the basic physical and chemical properties of polymethyl methacrylate ("Perspex" type) plastic. Thermal, elastic, crazing, solvent absorption, and mechanical properties are included and the effect of these on the service efficiency of a plastic structure is described. Experimental evidence is given concerning the essential role of tensile stress and absorbed solvent in causing crazing and recommendations concerning means to reduce or avoid the incidence of crazing are included. The basic thermal properties are compared with those of metals and the dangers of differential expansion in combined metal-plastic structures are noted, together with the serious effects of chilling of plastic structures during the "hot-forming" operation. Details are given of appropriate heat treatments designed to remove casting and workshop strains without causing distortion.

N-27241*

Aeronautical Research Council (Gt. Brit.)
SOME PRELIMINARY MODEL EXPERIMENTS ON THE WHIRLING OF SHAFTS. E. Downham. 1953. 20p. diagrs., photos. (ARC R & M 2768; ARC 13,733. Formerly RAE Structures 82)

Experimental work is now being done to establish a basis for the solution of whirling problems on turbine and contrarotating shaft systems in the design stage. This report is concerned primarily with the degree of accuracy to be expected from experiments on models. In experiments here described results are obtained for a simple cantilever system which are in close agreement with theory. With more complicated systems the error is somewhat greater owing to practical effects not covered by theory, though still acceptable for most design purposes.

N-27242*

Aeronautical Research Council (Gt. Brit.)
FLOW PATTERNS IN THE WAKE OF AN OSCILLATING AEROFOIL. J. B. Bratt. 1953. 27p. diagrs., photos. (ARC R & M 2773. Formerly ARC 13,001; FM 1423; 0.864)

Part I of the report gives an account of experiments made with the National Physical Laboratory smoke generator to obtain smoke patterns in the wake of an airfoil performing a rolling oscillation in a wind stream. Calculations are given in part II relating to the smoke patterns produced by a nozzle in uniform motion relative to (a) an infinite row of equally spaced two-dimensional discrete vortices of alternate sign and (b) an infinite two-dimensional vortex sheet with sinusoidal distribution of strength. Comparison with certain of the smoke patterns discussed in part I suggests that the wake vorticity can be treated very closely as a system of discrete vortices, and this is supported by the consideration that the elements of a continuous vortex sheet would in general be subject to normal induced velocities, which would tend to break the sheet up. The tendency for this to occur would be greater for the higher values of ω, wince the variation of vorticity with distance along the wake would be greater. The induced velocities for a uniform infinite vortex sheet would be zero.

N-27250*

Aeronautical Research Council (Gt. Brit.)
EXPERIMENTS ON RIGID PARACHUTE MODELS. R. Jones. 1953. 21p. diagrs., 10 tabs. (ARC R & M 2520. Formerly ARC 7218; Ae. 2363)

The investigations described in this report were undertaken for the Aeronautical Research Committee in order to study the effect of porosity on the stability of a parachute. Experiments were conducted to obtain data which could be applied in the usual way to the standard equations of motion.

N-27251*

Aeronautical Research Council (Gt. Brit.)
THE BUCKLING IN COMPRESSION OF PANELS WITH SQUARE TOP-HAT SECTION STRINGERS. W. S. Hemp and K. H. Griffin. 1953. 14p. diagrs., tab. (ARC R & M 2635; ARC 12,462. Formerly College of Aeronautics, Cranfield, Rept. 29)

A simplified panel model is described, together with a number of assumptions about the mode of its buckling. The approach to the calculation of the buckling stress is by splitting the panel into a number of flat plates and treating these by the ordinary plate theory. Use of the boundary conditions between these plates leads to a relation between the buckling stress and the variables of the panel geometry. The results thus obtained are compared with two sets of recent experimental work, and an appendix is included to show the effect of initial panel irregularities on the experimental determination of buckling stresses.

N-27252*

Aeronautical Research Council (Gt. Brit.)
INVESTIGATION OF SKIN BUCKLING. D. J. Farrar. 1953. 62p. diagrs., 12 tabs. (ARC R & M 2652; ARC 11,036; ARC 11,037; ARC 11,038. Formerly MOS S & TM 12A/47; S & TM 12B/47; S & TM 12C/47; Bristol Aeroplane Co., Ltd. CR 434)

The present tests were conducted on aluminum alloy plates in endwise compression, with varying conditions of edge support, to provide data on the buckling stress and post-buckling behavior of aircraft skins. All the plates tested were 35 inches long and nominally 0.064 inch thick. The plate width between the supports was varied between 35 and 120 times its thickness. Both clad (D.T.D. 546) and unclad (D.T.D. 646) material were tested. Three types of edge support were used: rows of steel balls in vee-grooved blocks, intended to imitate pin-edged conditions; rows of steel rollers in recessed blocks, intended to imitate clamp-edged conditions; and a single type of stringer used in previous panel tests. Measurements were made of the plate load and mean strain, and of the shape of the skin buckles. The test technique is discussed and the experimental results compared with theory.

N-27253*

Aeronautical Research Council (Gt. Brit.)
A REVISED INDEX OF MATHEMATICAL TABLES FOR COMPRESSIBLE FLOW. R. C. Tomlinson. 1953. 12p. diagrs. (ARC R & M 2691; ARC 11,582. Revised edition of ARC 9893; FM 976)

This report contains information of all the revelant tables known to the author, which can be obtained by workers outside the establishment of origin. The purpose of the index is to make available to workers in the field of compressible flow a reference from which they may trace a tabulation of any function they require, if it exists.

N-27254*

Aeronautical Research Council (Gt. Brit.)
WIND-TUNNEL TESTS ON A 12-FT DIAMETER HELICOPTER ROTOR. H. B. Squire, R. A. Fail and R. C. W. Eyre. 1953. 19p. diagrs., photos., 5 tabs. (ARC R & M 2695; ARC 12,524. Formerly RAE Aero 2324)

Measurements of the thrust, torque, and flapping angle for a 12-foot diameter rotor over a range of blade angle, shaft inclination, and tip-speed ratio have been made to give information on the validity of the standard rotor theory and of the effect of stalling on the retreating blade. Good agreement with the theory was obtained over the normal operating range, using airfoil characteristics determined from the measurements in the static thrust condition. Stalling was found to be progressive in character showing first by an increase in torque and flapping angle and later by a fall in thrust, as compared with the calculated values.

N-27255*

Aeronautical Research Council (Gt. Brit.)
SOME RELATED OSCILLATION PROBLEMS. W. J. Duncan. 1953. 12p. diagrs. (ARC R & M 2707; ARC 12,406. Formerly College of Aeronautics, Cranfield, Rept. 27)

Two simple means for establishing a relation between a pair of oscillation problems are briefly discussed. In the first, the displacements are connected by use of a differential operator. The set of natural frequencies is identical for the two problems and results of interest are obtained when the transformed boundary conditions can be physically interpreted. In this manner, it is shown, for example, that a flywheel on a uniform shaft can be transformed into a flexible coupling and a mass carried on a uniform beam into a flexible hinge. In the second, the connection is established by use of the concept of mechanical admittance. Here the frequency equations are simply related but the frequencies are not.

N-27256*

Aeronautical Research Council (Gt. Brit.)
VELOCITY DISTRIBUTION ON UNTAPERED SHEARED AND SWEPT-BACK WINGS OF SMALL THICKNESS AND FINITE ASPECT RATIO AT ZERO INCIDENCE. S. Neumark and J. Collingbourne. 1953. 50p. diagrs. (ARC R & M 2717; ARC 12,491. Formerly RAE Aero 2316)

This report is a continuation of an earlier one and puts forward several new solutions of the problems of velocity distribution on finite or semi-infinite untapered wings, at zero incidence. The solutions are based on the first order method of sources and sinks, which is shown to be sufficiently accurate to deal with problems involving tips or kinks. The fundamental case considered is that of a semi-infinite sheared wing, and the theory is built up to embrace finite sheared and sweptback wings, straight wings being dealt with as special cases. Complete detailed solutions are given for wings with biconvex parabolic profile, and the problems involving arbitrary profiles are investigated in a more general way. The velocity distribution in the regions of kinks and tips differing considerably from that on an infinite sheared wing, some local positive or negative drag arises in these regions. The drag distribution is calculated for the case of a biconvex parabolic profile, and it is found that, while the total resultant drag must obviously be nil for a finite wing, it may differ from zero for infinite or semi-infinite wings. This "potential" drag is generally small.

N-27257*

Aeronautical Research Council (Gt. Brit.)
SCHLIEREN TESTS ON SOME CONVENTIONAL TURBINE CASCADES. J. A. Dunsby. 1953. 14p. diagrs., photos. (ARC R & M 2728; ARC 12,893. Formerly NGTE R. 56)

Schlieren tests on a series of conventional turbine cascades have shown that the variations in performance at high speed can be accounted for by shock-wave and boundary-layer interaction. The rise in loss coefficient sometimes encountered at outlet Mach numbers of 0.6 to 0.8 is shown to be due to the formation of a λ shock series on the upper surface of the blade, the subsequent fall in loss coefficient and increase in deflection as the outlet Mach number rises to unity being caused by the formation of a shock system at outlet which forces the separated part of the boundary layer back on to the blade surface. It is shown that a λ shock series may form on a boundary layer which is apparently turbulent. This has not been observed before.

N-27258*

Aeronautical Research Council (Gt. Brit.)
STRESSES IN BUILT-UP BEAMS DUE TO AN ABRUPT CHANGE IN SHEAR STRESS AT A LOADING STATION. J. Taylor. 1953. 18p. diagrs., photos. (ARC R & M 2775; ARC 12,737. Formerly RAE Structures 48)

Owing to the abrupt change in shear stress at loading sections of beams there is a concentration of direct stress in the outer fibers of the beam near the loading section. A method of calculating this concentration is described. The highest stress concentrations occur in short deep beams and are greater for wooden than metal beams. The method is applied to the spars of two wooden aircraft and stress concentrations 1.06 and 1.4 are found at the fuselage attachments. Strain measurements were made at positions on a wooden beam under load and the theoretical predictions verified.

N-27259*

Aeronautical Research Council (Gt. Brit.)
THE EVALUATION OF DOWNWASH AT LARGE SPANWISE DISTANCES FROM A VORTEX LATTICE. H. C. Garner. 1953. 8p. diagrs., 6 tabs. (ARC R & M 2808. Formerly ARC 13,624; Perf. 727; S & C 2472)

This note explains an improved numerical method of evaluating the contributions to the downwash at moderate or large spanwise distances from a vortex lattice. By allowing freedom of choice of the chordwise positions of the discrete vortices of the lattice, it is possible to select three definite chordwise positions and strengths of vortices at each of these positions dependent on the chordwise pressure distribution, so as to determine the downwash with good accuracy for three particular pressure distributions proportional to $\cot \frac{1}{2}\theta$, $\sin \theta$, and $\sin 2\theta$. The corresponding chordwise loading factors have also been evaluated for deflected flaps.

N-27260*

Aeronautical Research Council (Gt. Brit.)
THE TWO-DIMENSIONAL SUBSONIC FLOW OF AN INVISCID FLUID ABOUT AN AEROFOIL OF ARBITRARY SHAPE. L. C. Woods. 1953. 58p. diagrs., 9 tabs. (ARC R & M 2811. Formerly ARC 13,240; FM 1456; ARC 13,395; FM 1475; ARC 13,460; FM 1489; ARC 13,533; FM 1499)

The paper describes and applies exact methods of calculating the incompressible flow about thick airfoils of general shape in a free stream, and about symmetrical airfoils between channel walls. One of these methods is extended to an approximate treatment of subsonic compressible flow by making use of von Kármán's transformation.

N-27263*

Aeronautical Research Council (Gt. Brit.)
GROWTH OF THE TURBULENT WAKE CLOSE BEHIND AN AEROFOIL AT INCIDENCE. D. A. Spence. 1953. 18p. diagrs. (ARC CP 125)

Knowledge of the variation close behind a trailing edge of the wake displacement thickness $\delta^* = \int \left(1 - \frac{u}{U}\right) dy$ is necessary in calculations of the circulation round an airfoil. Examination of data now available reveals that in the wake: (1) velocity profiles on either side of the line of minimum velocity may be derived from the corresponding trailing-edge boundary-layer profiles by change of scale of each coordinate; (2) the velocity defect at corresponding points follows a universal recovery law of the form $K(x - x_0)^{-\frac{1}{2}}$ even close to the trailing edge. An immediate consequence of these two empirical properties is a simple relation for the form parameter $H = \delta^*/\theta$ at points in the wake in terms of trailing-edge values. In conjunction with the momentum equation, this makes δ^* determinate. Agreement with experiment is very satisfactory.

N-27264*

Aeronautical Research Council (Gt. Brit.)
NOTE ON THE LIFT SLOPE, AND SOME OTHER PROPERTIES, OF DELTA AND SWEPT-BACK WINGS. E. F. Relf. 1953. 9p. diagrs., tab. (ARC CP 127)

In studying and comparing various theories for the determination of the distribution of loading on wings, Garner has given values for the lift slope of several families of sweptback and delta wings deduced from several different lifting-surface theories. In figure 8 of reference 1, Garner has plotted these lift slopes as functions of the aspect ratio A, for different values of the angle of sweep. It occurred to the writer to try plotting the ratio of the lift slope to that for elliptic loading instead of the lift slope itself, and when this was done it was noticed that the above ratio was very nearly independent of aspect ratio A, and gave a unique curve for all the available results when plotted against sweepback angle Λ. It has been possible to make some comparisons of theoretical deductions drawn from the above with measurements made in the C. A. T. on four delta wings, one swept wing, a tapered swept wing on a body, and two untapered swept wings on a body at high Reynolds numbers, where one could expect a close approximation to potential theory. While analyzing the C. A. T. results, values of the quantity K in the formula $C_D = C_{D_0} - \frac{K}{\pi A} C_L{}^2$ were collected and studied. Lastly, a study was made of scale effects in the C. A. T. tests.

N-27265*

Aeronautical Research Council (Gt. Brit.)
METHOD FOR THE DETERMINATION OF THE PRESSURE DISTRIBUTION OVER A FINITE THIN WING AT A STEADY LOW SPEED. G. J. Hancock. 1953. 11p. diagrs. (ARC CP 128)

For any given pressure distribution across a finite thin wing at low speed, the wing surface can be obtained by direct double integration. Therefore, the pressure distribution across a given wing surface may be obtained by the superposition of a number of solutions in which the wing surface is known for a prescribed pressure distribution. The method has been applied for the determination of the pressure distribution across a thin uncambered delta wing.

N-27266*

National Gas Turbine Establishment (Gt. Brit.)
AN ANALYSIS OF THE AIR FLOW THROUGH THE NOZZLE BLADES OF A SINGLE STAGE TURBINE. I. H. Johnston. 1953. 18p. diagrs. (ARC CP 131)

This memorandum presents the results of detailed traverses made on three of the nozzle assemblies designed for a single stage experimental turbine. The effects of pitch/chord ratio on gas outlet angle and total head loss are recorded and discussed in the light of corresponding work published elsewhere. The value of pitch/chord ratio giving minimum total head loss is found to compare well with the optimum pitching given by two-dimensional results obtained from cascade tests on blades of a similar nature.

N-27267*

Aeronautical Research Council (Gt. Brit.)
TESTS ON AN AXIAL COMPRESSOR WITH VARIOUS STATOR BLADE STAGGERS. R. A. Jeffs, E. L. Hartley and P. Rooker. 1953. 18p. diagrs., tab. (ARC CP 132)

This memorandum presents the results of a series of low speed tests on six stages of a medium stagger free vortex design of axial compressor blading, in which the stagger of the stator blades was varied over a wide range while the rotor blade stagger remained at its design figure. It is shown that efficiencies in excess of 85 percent were achieved over a range of stator blade stagger from -50° to +10°, compared with the design figure of -25.4°. This performance augurs well for the improvement of the performance of axial flow compressors away from their design point by the method of altering the stator blade stagger in some of the stages.

N-27268*

Aeronautical Research Council (Gt. Brit.)
A RELAXATION TREATMENT OF SHOCK WAVES.
L. C. Woods. 1953. 9p. diagrs. (ARC CP 134)

Emmons has given a relaxation method of dealing with shock waves when the compressible stream function is the dependent variable. This paper briefly outlines a procedure to adopt when $\log \frac{1}{q}$ (q = velocity magnitude) is taken as the dependent variable. A method of allowing for the presence of vorticity behind the shock wave is also given.

N-27287*

Ministry of Supply (Gt. Brit.)
THE STOVE-ENAMELLING OF ALUMINIUM ALLOYS - EFFECTS ON TENSILE PROPERTIES.
October 1953. 7p. diagrs. (MOS S & TM 3/53)

An investigation on the effects of short period heating of aluminum alloys in the range 120° C-200° C was made. Curves are presented showing the effects of heating on the 0.1 percent proof stress and ultimate stress of the material. A further investigation was made on the possibility of recovery of properties taking place either by prolonging the low-temperature stoving time or by leaving the material for a long time at room temperature after stoving.

N-27293*

Royal Aircraft Establishment (Gt. Brit.)
THE R.A.E. 4 FT x 3 FT EXPERIMENTAL LOW TURBULENCE WIND TUNNEL. PART IV. FURTHER TURBULENCE MEASUREMENTS.
H. Schuh. June 1953. 78p. diagrs., 7 tabs. (RAE Aero 2494)

Further measurements were made in the working section of this 4-foot by 3-foot wind tunnel. The influence of the number of screens in the bulge on the turbulence in the working section was studied, and the intensity and scale of turbulence was measured at the end of the second diffuser and at various places downstream and in the bulge.

N-27299*

Forest Products Research Lab. (Gt. Brit.)
TRIALS OF TIMBER FOR PLYWOOD MANUFACTURE. SMALL SCALE TESTS ON FOUR SPECIES (CANARIUM, OKWEN, SEPETIR AND SERRETTE). PROGRESS REPORT TWENTY-TWO. October 1953. 16p. 3 tabs. (Forest Products Research Lab.)

Four species of wood were evaluated as to their usefulness in the manufacture of plywood. Data on their processing, drying, gluing qualities, and pressure compression properties is given.

DECLASSIFIED NACA REPORTS

THE FOLLOWING REPORTS HAVE BEEN DECLASSIFIED FROM RESTRICTED TO UNCLASSIFIED, 12/7/53.

NACA RM A51E15

THE USE OF TWO-DIMENSIONAL SECTION DATA TO ESTIMATE THE LOW-SPEED WING LIFT COEFFICIENT AT WHICH SECTION STALL FIRST APPEARS ON A SWEPT WING. Ralph L. Maki. July 1951. 37p. diagrs., photo., 4 tabs. (NACA RM A51E15)

A procedure for estimating the wing lift coefficient for and the spanwise location of the first occurrence of maximum section lift on sweptback wings is presented. The method utilizes simplified lifting-surface theory, two-dimensional data, and simple sweep theory. The procedure is applied to a swept-wing model with and without trailing-edge split flaps and with several wing modifications. The predicted wing lift coefficients for the onset of stall are compared with measured values at which marked changes in force and moment characteristics occurred.

NACA RM A51I20

ARRANGEMENT OF BODIES OF REVOLUTION IN SUPERSONIC FLOW TO REDUCE WAVE DRAG. Morris D. Friedman. December 1951. 17p. diagrs. (NACA RM A51I20)

The effect of interference on the wave drag of a combination of bodies of revolution at zero angle of attack at supersonic speeds is investigated. Numerical calculations of the drag change obtainable from interference are carried out and curves are drawn for cases of two bodies of identical fineness ratio but with lengths in the ratio 2 to 1. Also considered is the special case of a three body combination with bilateral symmetry, for which it is found that the total wave drag can be 35 percent less than the total wave drag of the same bodies without interaction.

NACA RM A52A14b

EXPERIMENTAL INVESTIGATION OF THE DRAG OF 30°, 60°, AND 90° CONE CYLINDERS AT MACH NUMBERS BETWEEN 1.5 AND 8.2. Alvin Seiff and Simon C. Sommer. April 1952. 25p. photos., diagrs. (NACA RM A52A14b)

Free-flight drag measurements of 30°, 60°, and 90° cone cylinders, with cylinder fineness ratios of 1.2, at Mach numbers between 1.5 and 8.2, with Reynolds numbers in the order of 1 million, are presented. It is concluded that the Taylor and Maccoll theory for wave drag of cones is accurate for 60° cones at Mach numbers from 2 to 8 in the absence of gaseous imperfections. The base drag of the 60° cone cylinder is calculated from the experimental results at Mach numbers 2.0 to 4.5. Discontinuities along a streamline in the flow about and behind the models are observed and explained.

NACA RM A52A18

INSTRUMENTATION OF THE AMES SUPERSONIC FREE-FLIGHT WIND TUNNEL. Robert O. Briggs, William J. Kerwin and Stanley F. Schmidt. April 1952. 46p. photos., diagrs., 3 tabs. (NACA RM A52A18)

A description of the instruments used in the Ames supersonic free-flight wind tunnel to obtain a time-distance record of free-flying models over a 15-foot flight path is presented. The instruments include a chronograph and shadowgraph stations. The measurements are accurate to within 0.1 microsecond and 0.003 inch. Circuit diagrams are included.

NACA RM A52A24

THE AMES SUPERSONIC FREE-FLIGHT WIND TUNNEL. Alvin Seiff, Carlton S. James, Thomas N. Canning and Alfred G. Boissevain. April 1952. 30p. diagrs., photos. (NACA RM A52A24)

Research models are fired from a gun through the wind-tunnel test section in a direction opposite to the air stream which has a moderate supersonic Mach number. A wide range of Mach numbers, from low supersonic speeds up to M = 10, can be obtained. The equipment and test techniques are described. The methods used to measure drag, lift-curve slope, center of pressure, and damping in roll are given. The imperfections in the air stream and their effect on model tests are discussed.

NACA RM A52D22

THE FORCES AND PRESSURE DISTRIBUTION AT SUBSONIC SPEEDS ON A CAMBERED AND TWISTED WING HAVING 45° OF SWEEPBACK, AN ASPECT RATIO OF 3, AND A TAPER RATIO OF 0.5. Frederick W. Boltz and Carl D. Kolbe. July 1952. 166p. diagrs., 22 tabs. (NACA RM A52D22)

Lift, drag, pitching-moment, and pressure data for a model of a 45° sweptback cambered and twisted wing having an aspect ratio of 3 and a taper ratio of 0.5 are reported. The airfoil sections were the NACA 64A410 in planes inclined 45° to the plane of symmetry. Data are presented for Reynolds numbers from 4,000,000 to 18,000,000 at a Mach number of 0.25, for Reynolds numbers from 4,000,000 to 8,000,000 at a Mach number of 0.60, and for Mach numbers from 0.08 to 0.96 at a Reynolds number of 4,000,000. A comparison of the data with those for a plane wing of identical plan form is made. Pressure data at seven spanwise stations on the model are presented in tabular form.

NACA RM E50E05

ANALYTICAL INVESTIGATION OF TURBINES WITH ADJUSTABLE STATOR BLADES AND EFFECT OF THESE TURBINES ON JET-ENGINE PERFORMANCE. David H. Silvern and William R. Slivka. July 17, 1950. 51p. diagrs. (NACA RM E50E05)

Adjustable-stator turbines are applied to turbojet engines and probable performance is compared with conventional engines with and without variable-area exhaust nozzles. Variation in stator-exit angle and exhaust area was not excessive for wide ranges of engine output. Variation in turbine efficiency for contemporary turbines equipped with adjustable stators was small. Improvements from 4.5 to 17 percent in overall engine specific fuel consumption over conventional engines and from 2 to 8.5 percent over engines equipped with only adjustable-area exhaust nozzles were obtained at 60-percent rated power with adjustable-stator turbines and variable-area exhaust nozzles. The improvements depend on design parameters.

NACA RM E51F25

PRELIMINARY INVESTIGATION OF THE SUPERSONIC FLOW FIELD DOWNSTREAM OF WIRE-MESH NOZZLES IN A CONSTANT-AREA DUCT. Lawrence I. Gould. August 1951. 22p. diagrs., photos. (NACA RM E51F25)

An investigation was conducted in a 3.4- by 3.4-inch duct to determine the characteristics of the supersonic flow downstream of four wire-mesh screen nozzles with nominal design Mach numbers in the range between 1.97 and 2.58. Two types of disturbance were observed in the flow field: a fine network of interacting expansion and compression waves which formed immediately downstream of the screens and appeared to dissipate within 25 to 40 wave intersections; and relatively strong oblique shock waves that originated at the junctions of the screens and the walls and were reflected throughout the length of the duct. Regions of fairly uniform flow were found to exist. The total-pressure loss across the screens varied from 22 percent at Mach number 1.58 to 43 percent at Mach number 2.06.

NACA RM E51G27

SECONDARY FLOWS IN ANNULAR CASCADES AND EFFECTS ON FLOW IN INLET GUIDE VANES. Seymour Lieblein and Richard H. Ackley. August 1951. 63p. diagrs., tab. (NACA RM E51G27)

Qualitative discussion is presented of general nature of secondary flows in stationary annular cascades with thin wall boundary layers and radial design variation of circulation. Deviations from ideal mean outlet flows (based on blade-element performance) exist in potential-flow region of vanes because of conditions imposed by end-wall boundaries, displacement of wall boundary layers toward blade suction surfaces, and irrotationality requirement. As a consequence of existence of nonuniform radial flow across blade spacing, it may not generally be possible to obtain any arbitrarily specified design variation of turning angle along the radial height of a blade row. Quantitative turning angle corrections due to effects of secondary flows in axial-flow compressor inlet guide vanes were obtained from induced deflections of a superimposed vortex system in conjunction with an empirically determined correlation factor.

NACA RM E51H06

ANALYTICAL INVESTIGATION OF FLOW THROUGH HIGH-SPEED MIXED-FLOW TURBINE. Warner L. Stewart. October 1951. 22p. diagrs., photo. (NACA RM E51H06)

An analysis was made of the flow through a high-speed mixed-flow turbine based on axially symmetric conditions. The calculated weight flow was lower than that used in the two-dimensional design. The rotor could be redesigned to increase the weight flow and alleviate undesirable pressure gradients at the expense of increased rotor blade stresses.

NACA RM E52A09

EXPERIMENTAL INVESTIGATION OF FLOW IN AN ANNULAR CASCADE OF TURBINE NOZZLE BLADES OF CONSTANT DISCHARGE ANGLE. Milton G. Kofskey, Harold E. Rohlik and Daniel E. Monroe. March 1952. 28p. diagrs., photos. (NACA RM E52A09)

The experimental performance of turbine nozzle blades designed for a constant discharge angle was investigated at discharge hub Mach numbers of 1.18, 1.31, and 1.41. Flow characteristics are presented in terms of energy losses, angle gradients, and secondary flow effects. Blade efficiency decreased from 0.983 to 0.978 with increasing Mach number in the range investigated while angle variations in the loss regions became very large, indicating poorer blade performance than efficiency implies.

NACA RM E52D07

ANALYSIS OF STAGE MATCHING AND OFF-DESIGN PERFORMANCE OF MULTISTAGE AXIAL-FLOW COMPRESSORS. Harold B. Finger and James F. Dugan, Jr. June 1952. 38p. diagrs. (NACA RM E52D07)

An analysis is presented to give a qualitative picture of the operation of each stage in a high-pressure-ratio multistage compressor over a full range of operating flows and speeds and to point out methods of improving off-design performance. Single-stage performance results have been "stacked" to form a multistage compressor in which the design or match point of each stage has been arbitrarily selected. The effects of single-stage performance, stage match point, and stator-blade-angle adjustment are considered.

NACA RM E53A15

DESIGN CONSIDERATIONS FOR MIXED-FLOW CENTRIFUGAL COMPRESSORS WITH HIGH WEIGHT-FLOW RATES PER UNIT FRONTAL AREA. John D. Stanitz. March 1953. 48p. diagrs. (NACA RM E53A15)

An analysis is made of the factors affecting the weight-flow rate per unit frontal area of centrifugal compressors with axial-flow vaned diffusers preceded by mixed-flow vaneless sections. It is shown that, for specified inlet conditions to the impeller and vaned diffuser, the weight-flow rate is increased at the expense of pressure ratio and vice versa. Charts are presented to help the designer make a satisfactory compromise between weight-flow rate and pressure ratio. Some conclusions of the investigation are: (1) Prewhirl is of negligible value in centrifugal compressors designed for high weight-flow rates. (2) Transonic inlet flow conditions are desirable for high values of compressor weight-flow rate and pressure ratio. (3) At the inducer tip a value of 60° for the inlet relative flow angle, measured from the axial direction, results in approximately maximum values of compressor weight-flow rate per unit frontal area.

NACA RM L8H04

AERODYNAMIC CHARACTERISTICS OF TWO ALL-MOVABLE WINGS TESTED IN THE PRESENCE OF A FUSELAGE AT A MACH NUMBER OF 1.9. D. William Conner. October 28, 1948. 20p. diagrs., photos., tab. (NACA RM L8H04)

Half-span models of two wings of different plan form were tested both as all-movable surfaces and as fixed surfaces in the presence of a half fuselage in the Langley 9- by 12-inch supersonic blowdown tunnel at a Mach number of 1.9. One wing had a half-delta plan form with 60° leading-edge sweep and was tested at a Reynolds number of 1.9×10^6. The other wing had a rectangular plan form modified by an Ackeret type tip and was tested at a Reynolds number of 1.4×10^6.

NACA RM L8H12

YAW CHARACTERISTICS OF A 52° SWEPTBACK WING OF NACA 64_1-112 SECTION WITH A FUSELAGE AND WITH LEADING-EDGE AND SPLIT FLAPS AT REYNOLDS NUMBERS FROM 1.93×10^6 TO 6.00×10^6. Reino J. Salmi. November 8, 1948. 33p. diagrs., photos. (NACA RM L8H12)

Contains results of low-speed, high Reynolds number, wind-tunnel tests of a 52° sweptback wing in yaw. The effects of Reynolds number, flap deflection, and fuselage position on the static-lateral-stability parameters are given. Includes data at high yaw angles and effects of fuselage position on the sidewash characteristics in the region of a vertical tail.

NACA RM L8K04

LANGLEY FREE-FLIGHT-TUNNEL INVESTIGATION OF THE AUTOMATIC LATERAL STABILITY CHARACTERISTICS OF A MODEL EQUIPPED WITH A GYRO STABILIZING UNIT THAT PROVIDED EITHER FLICKER-TYPE OR HUNTING CONTROL. Robert O. Schade. January 11, 1949. 39p. diagrs., photos., 2 tabs. (NACA RM L8K04)

An investigation has been made in the Langley free-flight tunnel to determine the lateral stability of a flying model equipped with a gyro stabilizing unit which applied control in response to bank and yaw. The results are presented in the form of time histories of motions of the flying model with flicker and hunting type of control. A systematic calibration was made, and formulas were developed to determine the response of the gyro unit to angles of bank and yaw for various angles of cant and tilt.

NACA RM L9B18

LOW-SPEED WIND-TUNNEL INVESTIGATION OF THE LONGITUDINAL STABILITY CHARACTERISTICS OF A MODEL EQUIPPED WITH A VARIABLE-SWEEP WING. Charles J. Donlan and William C. Sleeman, Jr. May 23, 1949. 43p. tab., diagrs., photos. (NACA RM L9B18)

A wind-tunnel investigation of a variable sweep, complete model was conducted at angles of sweepback of 45°, 30°, 15°, and 0°. The investigation included the effect of various wing cutouts, a sharp-leading-edge wing section, a wing vane, flap deflection, and vertical location of the horizontal tail. The data permit an estimation of the amount of longitudinal wing translation required to compensate for the stability changes accompanying the change in sweep angle.

NACA RM L9E02

THE EFFECT OF SPAN AND DEFLECTION OF SPLIT FLAPS AND LEADING-EDGE ROUGHNESS ON THE LONGITUDINAL STABILITY AND GLIDING CHARACTERISTICS OF A 42° SWEPTBACK WING EQUIPPED WITH LEADING-EDGE FLAPS. George L. Pratt and Thomas V. Bollech. June 21, 1949. 26p. diagrs., photo. (NACA RM L9E02)

The effect of half-span and full-span split flaps through a deflection range of 0° to 60° on the low-speed longitudinal and power-off gliding characteristics of a wing with 42° sweepback at the leading edge equipped with leading-edge flaps and the effect of leading-edge roughness on the longitudinal stability of the wing equipped with leading-edge flaps is presented. The split-flap tests were made at a Reynolds number of 6.8×10^6, and the effect of leading-edge roughness on the wing equipped with leading-edge flaps was determined at Reynolds numbers of 3.0×10^6 and 4.7×10^6.

NACA RM L9E24

INVESTIGATION OF LOW-SPEED AILERON CONTROL CHARACTERISTICS AT A REYNOLDS NUMBER OF 6,800,000 OF A WING WITH LEADING EDGE SWEPT BACK 42° WITH AND WITHOUT HIGH-LIFT DEVICES. Thomas V. Bollech and George L. Pratt. July 19, 1949. 31p. diagrs., photo. (NACA RM L9E24)

Presents the lateral control, hinge-moment, aileron load, and balance-chamber-pressure characteristics of an aileron on a wing with leading edge swept back 42° with and without high-lift and stall-control devices. The investigation was carried out at a Reynolds number of 6,800,000 for an angle-of-attack range from -4° through the stall.

NACA RM L9H18a

MAXIMUM LIFT AND LONGITUDINAL STABILITY CHARACTERISTICS AT REYNOLDS NUMBERS UP TO 7.8 x 10^6 OF A 35° SWEPTFORWARD WING EQUIPPED WITH HIGH-LIFT AND STALL-CONTROL DEVICES, FUSELAGE, AND HORIZONTAL TAIL. Albert P. Martina and Owen J. Deters. February 9, 1950. 70p. diagrs., photo., 4 tabs. (NACA RM L9H18a)

An investigation was made in the Langley 19-foot pressure tunnel of a wing incorporating NACA 65-210 airfoil sections and having an aspect ratio of 5.8. High-lift and stall-control devices included split, single, and double slotted trailing-edge flaps; slats, extensible-, and drooped-nose flaps. Static longitudinal stability characteristics with midwing fuselage and several vertical locations of a horizontal tail were determined with flaps neutral and deflected. Most of the data are presented for a Reynolds number of 6.5 x 10^6 corresponding to a Mach number of 0.2. Lift, drag, and pitching-moment data are presented for all configurations. Some stall diagrams are presented.

NACA RM L9L20a

EFFECTS OF PLAIN AND STEP SPOILER LOCATION AND PROJECTION ON THE LATERAL CONTROL CHARACTERISTICS OF A PLAIN AND FLAPPED 42° SWEPTBACK WING AT A REYNOLDS NUMBER OF 6.8 x 10^6. Thomas V. Bollech and George L. Pratt. February 14, 1950. 43p. diagrs., photos. (NACA RM L9L20a)

Presents the effects of spoiler geometry and location on the low-speed lateral-control characteristics of a wing swept back 42° at the leading edge with and without high-lift and stall-control devices. The investigation was carried out at a Reynolds number of 6.8 x 10^6 and through an angle-of-attack range from -4° through the stall.

NACA RM L50A04

LONGITUDINAL CHARACTERISTICS OF TWO 47.7° SWEPTBACK WINGS WITH ASPECT RATIOS OF 5.1 AND 6.0 AT REYNOLDS NUMBERS UP TO 10 x 10^6. Reino J. Salmi and Robert J. Carros. March 30, 1950. 25p. diagrs., photo. (NACA RM L50A04)

Contains results of low-speed wind-tunnel tests on two wings of 47.7° sweepback and aspect ratios of 5.1 and 6.0. Aerodynamic data are presented for a range of Reynolds numbers from about 1.1 x 10^6 to 10.0 x 10^6. Results show that abrupt unstable pitching moments occur at moderate lift coefficients. The maximum lift coefficients are about 1.20 and are only slightly affected by Reynolds numbers, but a considerable scale effect occurred for the lift coefficient at which the pitching moment broke. A maximum lift-drag ratio of about 27.8 was obtained with the aspect ratio 6.0 wing.

NACA RM L50F20

EFFECTS OF LEADING-EDGE DEVICES AND TRAILING-EDGE FLAPS ON LONGITUDINAL CHARACTERISTICS OF TWO 47.7° SWEPTBACK WINGS OF ASPECT RATIOS 5.1 AND 6.0 AT A REYNOLDS NUMBER OF 6.0 x 10^6. Reino J. Salmi. August 30, 1950. 105p. diagrs., photos., tab. (NACA RM L50F20)

Contains results of wind-tunnel tests at a Reynolds number of 6.0 x 10^6 (Mach number of 0.14) on the effects of leading-edge and trailing-edge flaps on the longitudinal stability characteristics of two 47.7° sweptback wings of aspect ratios 5.1 and 6.0. The effects of roughness, fuselage interference, and wing fences are also presented.

NACA RM L51B27

CONSIDERATIONS ON A LARGE HYDRAULIC JET CATAPULT. Upshur T. Joyner and Walter B. Horne. April 12, 1951. 57p. diagrs., photos., tab. (NACA RM L51B27)

A survey of various types of catapults, which has been made in connection with the problem of accelerating a large (100,000 lb) car along a track to a speed of 150 miles per hour, is given. A hydraulic jet catapult is indicated as the best-suited among these catapult types for the purpose intended, and various design problems of this type are treated. Equations are given for calculating the performance of the jet and of the test car, and consideration is given to the physical conditions affecting the jet flow. Design procedures are presented for the jet nozzle and for the bucket on the car which receives the jet and imparts thrust to the car. The expected propulsive efficiency of the jet catapult is given and the effect of a side wind on the jet trajectory is calculated.

NACA RM L51C20

LOW-SPEED STATIC LONGITUDINAL AND LATERAL STABILITY CHARACTERISTICS OF TWO LOW-ASPECT-RATIO WINGS CAMBERED AND TWISTED TO PROVIDE A UNIFORM LOAD AT A SUPERSONIC FLIGHT CONDITION. Lewis R. Fisher. June 6, 1951. 24p. diagrs., photos. (NACA RM L51C20)

A delta wing and a tapered sweptback wing of aspect ratios 1.56 and 2.00, respectively, both of which were cambered and twisted so as to provide a uniform load distribution for a supersonic flight condition, were tested in combination with a fuselage at Reynolds numbers between 384,000 and 1,550,000 in order to determine the low-speed lift-drag and static stability characteristics of such wings.

NACA RM L51E24

LOW-SPEED INVESTIGATION OF THE EFFECTS OF SINGLE SLOTTED AND DOUBLE SLOTTED FLAPS ON A 47.7° SWEPTBACK-WING - FUSELAGE COMBINATION AT A REYNOLDS NUMBER OF 6.0×10^6. Ernst F. Mollenberg and Stanley H. Spooner. September 1951. 23p. diagrs., photos., 3 tabs. (NACA RM L51E24)

The effects of deflection of partial-span single and double slotted flaps in combination with leading-edge flaps on the longitudinal aerodynamic characteristics of a 47.7° sweptback-wing - fuselage combination are shown. The wing had an aspect ratio of 5.1, a taper ratio of 0.383, and NACA 64-210 airfoil sections. The tests were conducted in the Langley 19-foot pressure tunnel at a Reynolds number of 6.0×10^6 and a Mach number of 0.14.

NACA RM L51H13

LOW-SPEED CHARACTERISTICS OF A 45° SWEPTBACK WING OF ASPECT RATIO 8 FROM PRESSURE DISTRIBUTIONS AND FORCE TESTS AT REYNOLDS NUMBERS FROM 1,500,000 TO 4,800,000. Robert R. Graham. October 1951. 54p. diagrs., photos., tab. (NACA RM L51H13)

Results of pressure-distribution tests at Reynolds numbers of 1,500,000 and 4,000,000 and of force tests through a Reynolds number range from 1,500,000 to 4,800,000 are presented. Pressure data for 4,000,000 Reynolds number are tabulated and data for both Reynolds numbers are presented as wing and section force and moment characteristics. Effects of a fence configuration and leading-edge roughness were investigated at a Reynolds number of 4,000,000.

NACA RM L51J04

LOW-SPEED LONGITUDINAL CHARACTERISTICS OF A 45° SWEPTBACK WING OF ASPECT RATIO 8 WITH HIGH-LIFT AND STALL-CONTROL DEVICES AT REYNOLDS NUMBERS FROM 1, 500, 000 TO 4, 800, 000. George L. Pratt and E. Rousseau Shields. February 1952. 76p. diagrs., photo., 2 tabs. (NACA RM L51J04)

The low-speed static longitudinal stability characteristics of a wing having 45° sweepback of the quarter-chord line, an aspect ratio of 8, a taper ratio of 0.45, and NACA 63_1A012 airfoil sections parallel to the air stream were determined in the Langley 19-foot pressure tunnel at Reynolds numbers from 1.5×10^6 to 4.8×10^6. The effects of combinations of leading-edge and trailing-edge flaps of various spans, upper-surface flow-control fences, and a fuselage on the longitudinal stability were investigated.

NACA RM L51J26

A PRELIMINARY LOW-SPEED WIND-TUNNEL INVESTIGATION OF A THIN DELTA WING EQUIPPED WITH A DOUBLE AND A SINGLE SLOTTED FLAP. Richard G. MacLeod. January 1952. 12p. diagrs., photo., 2 tabs. (NACA RM L51J26)

Contains results and discussion of a preliminary wind-tunnel investigation at low speeds of a thin delta wing equipped with a slotted flap. Results indicated that the maximum lift coefficient could be increased from 1.45 to 1.86 by deflecting the double slotted flap 50°. The angle of attack of the model necessary to obtain a given lift coefficient was considerably reduced by the addition of the slotted flaps.

NACA RM L52C11

LOW-SPEED LONGITUDINAL AERODYNAMIC CHARACTERISTICS OF A TWISTED AND CAMBERED WING OF 45° SWEEPBACK AND ASPECT RATIO 8 WITH AND WITHOUT HIGH-LIFT AND STALL-CONTROL DEVICES AND A FUSELAGE AT REYNOLDS NUMBERS FROM 1.5×10^6 TO 4.8×10^6. Reino J. Salmi. June 1952. 76p. diagrs., photo., 2 tabs. (NACA RM L52C11)

The low-speed static longitudinal stability characteristics of a twisted and cambered wing having 45° sweepback of the quarter-chord line, an aspect ratio of 8, a taper ratio of 0.45, and an NACA 63_1A012 thickness distribution parallel to the air stream, were investigated in the Langley 19-foot pressure tunnel at Reynolds numbers from 1.5×10^6 to 4.8×10^6. The effects of combinations of leading-edge and trailing-edge flaps of various spans, upper-surface flow-control fences, and a fuselage on the longitudinal stability were investigated.

NACA RM L52F04

LOW-SPEED INVESTIGATION OF THE EFFECTS OF NACELLES ON THE LONGITUDINAL AERODYNAMIC CHARACTERISTICS OF A 60° SWEPTBACK DELTA-WING - FUSELAGE COMBINATION WITH NACA 65A003 AIRFOIL SECTIONS. William I. Scallion. July 1952. 21p. diagrs., photos., 3 tabs. (NACA RM L52F04)

An investigation was made in the Langley full-scale tunnel to determine the effects of symmetrically located wing nacelles on the low-speed aerodynamic characteristics of a 60° sweptback delta-wing - fuselage combination. The wing section was an NACA 65A003 airfoil section and the aspect ratio was 2.31. The model was tested with nacelles at three chordwise locations at each of three spanwise stations. Lift, pitching moment, and drag were obtained through the angle-of-attack range from -3° to stall at zero yaw at Reynolds numbers from 1.55×10^6 to 2.77×10^6 and Mach numbers from 0.07 to 0.12.

NACA RM L52J03a

EFFECTS OF TWIST AND CAMBER ON THE LOW-SPEED LONGITUDINAL STABILITY CHARACTERISTICS OF A 45° SWEPTBACK WING OF ASPECT RATIO 8 AT REYNOLDS NUMBERS FROM 1.5×10^6 TO 4.8×10^6 AS DETERMINED BY PRESSURE DISTRIBUTIONS, FORCE TESTS, AND CALCULATIONS. George L. Pratt. December 1952. 104p. diagrs., photo., 3 tabs. (NACA RM L52J03a)

The low-speed longitudinal stability characteristics of a 45° sweptback wing of aspect ratio 8 having twist and cambered airfoil sections were investigated by means of force and pressure-distribution measurements at Reynolds numbers from 1.5×10^6 to 4.8×10^6 in the Langley 19-foot pressure tunnel. The effects of Reynolds number, leading-edge roughness, upper-surface fences, and leading-edge and trailing-edge flaps have been determined. A comparison has been made between the results obtained on the twisted and cambered wing and the results obtained on a wing of similar plan form without twist and having symmetrical airfoil sections of the same thickness distributions. The experimental pressure-distribution loadings have been compared to calculated loadings.

NACA RM L52K26

THE LOW-SPEED LIFT AND PITCHING-MOMENT CHARACTERISTICS OF A 45° SWEPTBACK WING OF ASPECT RATIO 8 WITH AND WITHOUT HIGH-LIFT AND STALL-CONTROL DEVICES AS DETERMINED FROM PRESSURE DISTRIBUTIONS AT A REYNOLDS NUMBER OF 4.0×10^6. Thomas V. Bollech and William M. Hadaway. January 1953. 57p. diagrs., photo. (NACA RM L52K26)

This paper presents the low-speed lift and pitching-moment characteristics of a 45° sweptback wing having an aspect ratio of 8, a taper ratio of 0.45, and incorporating NACA 63_1A012 airfoil sections in the streamwise direction with and without high-lift and stall-control devices as determined by pressure distribution at a Reynolds number of 4.0×10^6 and through an angle-of-attack range from -4° through the stall.

NACA RM L53A02

A THEORETICAL ANALYSIS OF THE DISTORTION OF FUEL-SPRAY-PARTICLE PATHS IN A HELICOPTER RAM-JET ENGINE DUE TO CENTRIFUGAL EFFECTS. S. Katzoff and Samuel L. Smith, III. April 1953. 44p. diagrs., tab. (NACA RM L53A02)

A theoretical analysis was made of the centrifugal effects on the fuel-spray paths in a rotating ram-jet engine tip-mounted on a helicopter blade. The differential equations of motion of the particles were set up and solved numerically for sixty-four selected combinations of the parameters. Plots of all the calculated paths are given. A simple method for determining the approximate final direction of the particles was developed. The results indicated that, because of the centrifugal action, the larger fuel particles were likely to hit the outer walls of the ram jet before burning, so that loss of thrust would result. Several methods of correcting for this effect are suggested.

NACA RM L53A07

COMPRESSIBLE-FLOW SOLUTIONS FOR THE ACTUATOR DISK. James B. Delano and John L. Crigler. March 1953. 70p. diagrs. (NACA RM L53A07)

Solutions for the actuator disk in subsonic compressible flow are presented. The induced flow phenomena for compressible and incompressible flow are shown to be widely different. Solutions have been obtained for flows with subsonic and supersonic wakes. Calculations are presented to show the large gain in efficiency that can be obtained by the use of disks in tandem for power loadings greater than those required to choke single disks.

NACA RM L53C20

AERODYNAMIC CHARACTERISTICS AT HIGH AND LOW SUBSONIC MACH NUMBERS OF FOUR NACA 6-SERIES AIRFOIL SECTIONS AT ANGLES OF ATTACK FROM -2° TO 31°. Homer B. Wilson, Jr. and Elmer A. Horton. June 1953. 48p. diagrs., photo., tab. (NACA RM L53C20)

A two-dimensional investigation of the NACA 64-006, 64-008, 64-010, and 64_1-012 airfoil sections has been made in the Langley low-turbulence pressure tunnel at angles of attack of -2° to 31° and Mach numbers of 0.3 to that for tunnel choke. Measurements were made of the lift, drag, and pitching-moment coefficients for the airfoil models in the smooth condition and with leading-edge roughness. One airfoil model, the NACA 64_1-012, was also tested with a roughness strip at the 20-percent-chord station.

NACA RM L53D02

INVESTIGATION OF A RAM-JET-POWERED HELICOPTER ROTOR ON THE LANGLEY HELICOPTER TEST TOWER. Paul J. Carpenter and Edward J. Radin. June 1953. 32p. diagrs., photos. (NACA RM L53D02)

A helicopter rotor powered by tip-located ram-jet engines has been investigated on the Langley helicopter test tower. The propulsive and aerodynamic characteristics of the isolated engines were studied also in the nonwhirling condition in a small wind tunnel. The basic hovering characteristics, as well as the aerodynamic and propulsive characteristics of the engines, have been obtained.

NACA RM L53E07

AN EXPLORATORY INVESTIGATION OF SOME TYPES OF AEROELASTIC INSTABILITY OF OPEN AND CLOSED BODIES OF REVOLUTION MOUNTED ON SLENDER STRUTS. S. A. Clevenson, E. Widmayer, Jr. and Franklin W. Diederich. June 1953. 44p. diagrs., photos., 3 tabs. (NACA RM L53E07)

The aeroelastic instability of rigid open and closed bodies of revolution mounted on thin, flexible struts has been investigated experimentally at low speeds. Three types of instability were observed - coupled flutter, divergence, and an uncoupled oscillatory instability which consists in continuous or intermittent small-amplitude yawing oscillations. An attempt has been made to calculate the airspeeds and, in the case of the oscillatory phenomena, the frequencies at which these types of instability occur by using slender-body theory for the aerodynamic forces on the bodies.

NACA RM L53H18a

COMPARISON OF THE PERFORMANCE OF A HELICOPTER-TYPE RAM-JET ENGINE UNDER VARIOUS CENTRIFUGAL LOADINGS. Edward J. Radin and Paul J. Carpenter. October 1953. 17p. diagrs., photos. (NACA RM L53H18a)

The effect of centrifugal loadings on the performance of a helicopter-type ram-jet engine has been determined on the Langley helicopter test tower. Halving the centrifugal loading by doubling the radius reduced the minimum specific fuel consumption at all ram-jet velocities and increased the maximum propulsive thrust by approximately 12.5 percent at a ram-jet-engine velocity of 630 fps.

NACA RM L53J07

GUST-TUNNEL INVESTIGATION OF THE EFFECT OF LEADING-EDGE SEPARATION ON THE NORMAL ACCELERATIONS EXPERIENCED BY A 45° SWEPTBACK-WING MODEL IN GUSTS. George L. Cahen. November 1953. 16p. diagrs., photos., tab. (NACA RM L53J07)

An investigation of the effect of leading-edge separation on the loads experienced by a 45° sweptback-wing model in gusts indicated that separation increased the load, the amount of load increase apparently depending upon the gust-gradient distance and velocity. It was further indicated that the rate of change in angle of attack due to the gust and the extent of penetration into the gust are both important in determining whether separation occurs.

THE FOLLOWING REPORTS HAVE BEEN DECLASSIFIED FROM RESTRICTED TO UNCLASSIFIED, 12/8/53.

NACA RM E8B19

ALTITUDE-WIND-TUNNEL INVESTIGATION OF A 3000-POUND-THRUST AXIAL-FLOW TURBOJET ENGINE. III - ANALYSIS OF COMBUSTION-CHAMBER PERFORMANCE. Carl E. Campbell. August 23, 1948. 47p. diagrs., photos. (NACA RM E8B19)

Combustion-chamber performance characteristics of a 3000-pound-thrust axial-flow turbojet engine have been determined from an investigation of the complete engine in the Cleveland altitude wind tunnel over a range of simulated altitudes and flight Mach numbers. The effect of variations in altitude and flight Mach number on combustion efficiency, combustion-chamber total-pressure losses, engine-cycle efficiency, and the fractional loss in engine-cycle efficiency resulting from combustion-chamber pressure losses is presented for various engine configurations.

NACA RM E8F09e

ALTITUDE-WIND-TUNNEL INVESTIGATION OF A 4000-POUND-THRUST AXIAL-FLOW TURBOJET ENGINE. VI - COMBUSTION-CHAMBER PERFORMANCE. I. Irving Pinkel and Harold Shames. August 4, 1948. 20p. diagrs., photo., tabs. (NACA RM E8F09e)

Performance of three types of combustion chamber operating in a 4000-pound-thrust axial-flow turbojet engine is presented for altitudes from 5000 to 40,000 feet and ram pressure ratios from 1.00 to 1.86. The loss in cycle efficiency due to the pressure losses in the combustion chamber was found to be of little consequence in the design operating range of the engine. Combustion efficiency improved with engine speed and ram pressure ratio at all altitudes and decreased with increasing altitude. At rated engine speed, the altitude effect on combustion efficiency was no greater than 5 percent.

NACA RM E9I22

EFFECT OF AIR DISTRIBUTION ON RADIAL TEMPERATURE DISTRIBUTION IN ONE-SIXTH SECTOR OF ANNULAR TURBOJET COMBUSTOR. Herman Mark and Eugene V. Zettle. April 5, 1950. 54p. diagrs. (NACA RM E9I22)

An investigation was conducted in a one-sixth segment of an annular turbojet combustor to determine a method of controlling radial exhaust gas temperature distribution. Two general methods were studied: (1) ducting the dilution air into the combustion chamber in a predetermined manner through hollow radial struts and (2) modifying the combustor basket-wall open-hole area. Results of the investigation indicated that: Modifications of the hollow radial struts had some effect on the exhaust-gas temperature distribution but for the combustor investigated complete control was impossible with this method. Secondary-zone basket-wall modifications, however, have a large effect on exhaust-gas temperature distribution provided the primary-zone basket wall is suitable.

NACA RM L50C23

DITCHING TESTS WITH A 1/16-SIZE MODEL OF THE NAVY XP2V-1 AIRPLANE AT THE LANGLEY TANK NO. 2 MONORAIL. Lloyd J. Fisher and Robert P. Tarshis. May 18, 1950. 40p. diagrs., photos., tab. (NACA RM L50C23)

Model investigations to determine the ditching characteristics of the Navy XP2V-1 airplane are described. Various landing configurations were simulated and the performance of the model was determined from visual observations, motion-picture records, and time-history accelerometer records.

The results of the investigation indicate that the airplane should be ditched at the normal landing attitude with the flaps fully extended. Extensive damage to the fuselage will occur and the airplane probably will dive. If a trapezoidal hydroflap 4 feet by 2 feet by 1 foot is attached to the airplane at station 192.4, diving will be prevented.

NACA RM L50K07

WIND-TUNNEL INVESTIGATION OF THE EFFECT OF CHORDWISE FENCES ON LONGITUDINAL STABILITY CHARACTERISTICS OF AN AIRPLANE MODEL WITH A 35° SWEPTBACK WING. M. J. Queijo and Byron M. Jaquet. December 18, 1950. 47p. diagrs., photos. (NACA RM L50K07)

An experimental investigation was made in the Langley stability tunnel to determine the effects of various chordwise fences on the longitudinal stability characteristics of an airplane model with a 35° sweptback wing. The investigation included the determination of the effects of fence shape, size, and position on the longitudinal characteristics of several model configurations.

NACA RM L51H17

WIND-TUNNEL INVESTIGATION OF THE EFFECTS OF HORIZONTAL-TAIL POSITION ON THE LOW-SPEED LONGITUDINAL STABILITY CHARACTERISTICS OF AN AIRPLANE MODEL WITH A 35° SWEPTBACK WING EQUIPPED WITH CHORDWISE FENCES. M. J. Queijo and Walter D. Wolhart. November 1951. 28p. diagrs., photo., tab. (NACA RM L51H17)

An experimental investigation was made in the Langley stability tunnel to determine whether the low-speed longitudinal stability characteristics of a model with a 35° sweptback wing could be improved appreciably by lowering the horizontal tail. The investigation included tests of several components of the model and of various model configurations with chordwise fences on the wing.

THE FOLLOWING REPORTS HAVE BEEN DECLASSIFIED FROM RESTRICTED TO UNCLASSIFIED, 12/10/53.

NACA RM A9B09

EFFECTS OF WING-TIP TURRETS ON THE AERODYNAMIC CHARACTERISTICS OF A TYPICAL BOMBER-WING MODEL. Lee E. Boddy and Fred B. Sutton. March 28, 1949. 22p. diagrs., photos. (NACA RM A9B09)

Results are presented of an experimental investigation of the effects of wing-tip gun turrets with various modifications upon the aerodynamic characteristics of a typical bomber-wing model. The data indicate that the addition of the turrets had negligible effects upon the lift and pitching-moment characteristics of the wing. The drag coefficient was increased by 0.005 up to a Mach number of 0.70, and the Mach number of drag divergence was decreased by 0.05.

NACA RM L8H19

WIND-TUNNEL TESTS OF A SWEPT-BLADE PROPELLER AND RELATED STRAIGHT BLADES HAVING THICKNESS RATIOS OF 5 AND 6 PERCENT. W. H. Gray. November 10, 1948. 63p. diagrs., photos., tab. (NACA RM L8H19)

Compares three related sets of blades, one conventional in design, one sweptback, and one embodying relatively thin sections. Actual design conditions could not be duplicated in the wind tunnel. The sweptback blades were in general inferior to the other two designs; the greatest difference in envelope efficiency was 2.5 percent. The contributing factors to the inferiority of the sweptback blades were insufficiently high Mach number to show the full effects of sweep and propeller operation away from the design conditions.

THE FOLLOWING REPORTS HAVE BEEN DECLASSIFIED FROM RESTRICTED TO UNCLASSIFIED, 12/11/53.

NACA RM E50E18a

SIMULATED ALTITUDE PERFORMANCE OF TWO ANNULAR COMBUSTORS WITH CONTINUOUS AXIAL OPENINGS FOR ADMISSION OF PRIMARY AIR. Eugene V. Zettle and Herman Mark. August 3, 1950. 39p. diagrs. (NACA RM E50E18a)

Methods of introducing and distributing air and fuel in turbojet-engine combustors were evaluated with two fuels, AN-F-32 and AN-F-58. Investigations were made with two single-annulus liners in a one-quarter sector of a 25-1/2 inch diameter turbojet combustor. Altitude performance data for these combustors are compared with existing data for a production-model double-annulus combustor. Altitude operating limits and combustion efficiencies of both single-annulus combustors were considerably higher than those of the double-annulus combustor; combustion efficiencies were insensitive to changes in fuel-air ratio.

NACA RM E52I17

EFFECT OF MAGNITUDE OF VIBRATORY LOAD SUPERIMPOSED ON MEAN TENSILE LOAD OF MECHANISM OF AND TIME TO FRACTURE OF SPECIMENS AND CORRELATION TO ENGINE BLADE. Robert R. Ferguson. November 1952. 26p. diagrs., photos., 4 tabs. (NACA RM E52I17)

Tensile fatigue tests were run on specimens of seven turbine blade alloys at a test temperature of 1500° F and a mean stress of 22,000 pounds per square inch with superimposed alternating stresses of 0, ±5000, ±10,000, and ±15,000 pounds per square inch. The same three types of fracture occurring in turbine blades - stress rupture, stress rupture followed by fatigue, and fatigue - were obtained in the specimens. The type of fracture obtained was found to be a function of the material and the magnitude of the alter-

nating stress. With increasing alternating stress, the mechanism of failure changed from stress rupture to stress rupture followed by fatigue and then to fatigue. Six of the seven alloys showed the same mechanism of failure for specimens tested at an alternating stress of ±5000 pounds per square inch as for blades tested in a J33-9 jet engine.

NACA RM E53G07

INVESTIGATION OF A CHROMIUM PLUS ALUMINUM OXIDE METAL-CERAMIC BODY FOR POSSIBLE GAS TURBINE BLADE APPLICATION. Charles A. Hoffman. November 1953. 12p. diagrs., photos., 4 tabs. (NACA RM E53G07)

A metal-ceramic composition containing approximately 80 percent chromium plus 20 percent aluminum oxide (Al_2O_3) by weight has been investigated for possible gas-turbine blade use. The results of modulus-or-rupture, thermal-shock, and blade-performance studies indicate that this material may have adequate thermal-shock resistance; however, the strength for this application appears marginal.

NACA RM L8D19

THEORETICAL ANALYSIS OF THE MOTIONS OF AN AIRCRAFT STABILIZED IN ROLL BY A DISPLACEMENT-RESPONSE, FLICKER-TYPE AUTOMATIC PILOT. Howard J. Curfman, Jr. and William N. Gardner. July 7, 1948. 33p. diagrs., tab. (NACA RM L8D19)

A general analysis is presented which allows the rolling motions of an aircraft using a displacement-response, flicker-type automatic pilot to be determined, and charts are included for finding the amplitude and period of steady-state oscillations of any aircraft. Current trends in pilotless-aircraft designs indicate that small amplitude residual oscillations are possible with the topic system. The analysis shows close agreement with roll-simulator tests.

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